NAVIGATION CONCEPTS FOR THE JAMES WEBB SPACE TELESCOPE

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ABSTRACT

This paper evaluates the performance that can be achieved using candidate ground and onboard navigation approaches for operation of the James Webb Space Telescope, which will be in an orbit about the Sun-Earth L2 libration point. The ground navigation approach processes standard range and Doppler measurements from the Deep Space Network. The onboard navigation approach processes celestial object measurements and/or ground-to-spacecraft Doppler measurements to autonomously estimate the spacecraft's position and velocity and Doppler reference frequency. Particular attention is given to assessing the absolute position and velocity accuracy that can be achieved in the presence of the frequent spacecraft reorientations and momentum unloads planned for this mission. The ground navigation approach provides stable navigation solutions using a tracking schedule of one 30-minute contact per day. The onboard navigation approach that uses only optical quality celestial object measurements provides stable autonomous navigation solutions. This study indicates that unmodeled changes in the solar radiation pressure cross-sectional area and unmodeled momentum unload velocity changes are the major error sources. These errors can be mitigated by modeling these changes, by estimating corrections to compensate for the changes, or by including acceleration measurements.

1 – INTRODUCTION

The James Webb Space Telescope (JWST), which is one of the next generation of space telescopes, is planned for launch in 2011. From a large halo orbit about the L2 Sun-Earth libration point (located 1.5 million kilometers (km) from the Earth in the anti-Sun direction), JWST will study the early evolution of the universe. The Mission Engineering and Systems Analysis Division at the Goddard Space Flight Center (GSFC) is supporting the JWST project by developing navigation concepts that meet nominal orbit determination accuracy requirements on the order of 50 km in position and 20 millimeters per second (mm/s) in velocity (3-sigma). These requirements are challenging because of the unusually large solar radiation pressure (SRP) forces that will be experienced by the spacecraft and the frequent attitude reorientations and unbalanced momentum unloads that are planned for this mission.

This paper evaluates the feasibility of the following approaches for meeting the navigation requirements:

- Ground navigation using standard range and/or Doppler measurements from the Deep Space Network (DSN), which is the current mission baseline
- Onboard navigation using the communications hardware and Sun sensor baselined for this mission
- Onboard navigation using optical celestial navigation sensors that can measure the angle between the Earth and the Moon or a star and the Moon.

Preliminary covariance analysis was performed to assess the sensitivity of the orbit determination accuracy to the frequency and magnitude of the momentum unloads. High-fidelity orbit determination simulations were performed to evaluate the candidate navigation approaches in more detail. Particular attention was given to assessing the position and velocity accuracy that can be achieved in the presence of the frequent spacecraft reorientations and momentum unloads planned for this mission, with and without the inclusion of accelerometer measurements.

2 –JWST SPACECRAFT CONFIGURATION

The JWST spacecraft has a unique design (see Figure 1 for deployed configuration). To protect the delicate optics of the telescope from direct sunlight, a 200 meter^2 (m^2) Sun shield separates the science instruments from the spacecraft bus. The telescope is always on the anti-Sun side of the Sun shield, with the spacecraft bus always on the sunward side. In order to point the telescope within its field of regard (FOR), the JWST spacecraft has a limited 68° pitch range and a 5° roll range. The spacecraft attitude is reoriented frequently (e.g. weekly) to change the telescope's field-of-view. The maximum change is about ± 30° with respect to the L2-to-Sun line. As a result, the spacecraft's cross-sectional area with respect to the Sun varies from about 173 m^2 to 200 m^2, with a mean of about 190 m^2. This variation causes up to a 5% variation in the SRP force acting on the spacecraft.

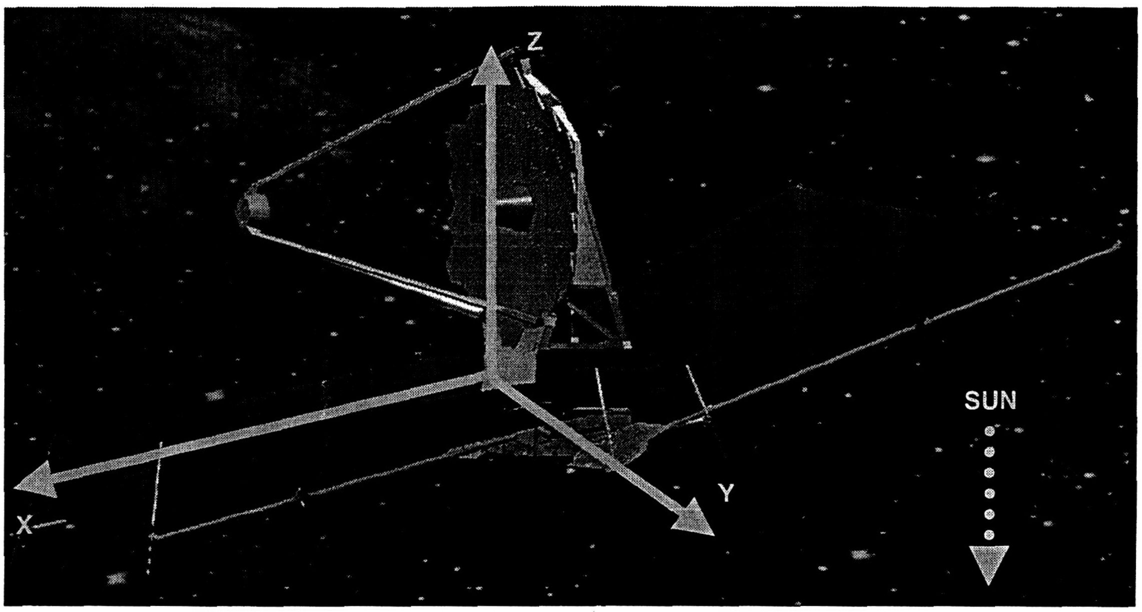

Figure 1. Northrup Grumman Space Technology JWST Design

As the telescope moves from target to target, solar torques build up within the momentum wheels. Figure 2, which shows the rate of momentum buildup for spacecraft attitudes within the FOR, indicates that the momentum buildup is particularly sensitive to the roll angle. The momentum control system consists of six momentum wheels with a total storage of 40 Newton-meter-seconds (N-m-s). The momentum is unloaded via the Reaction Control System (RCS) whenever the wheels become saturated. The RCS consists of four hydrazine 1 pound (lbf) thrusters, as shown in Figure 3. The orientation of the thrusters is restricted because of contamination issues with the Sun shield and science instruments and mass limitations. All four RCS thrusters are placed on the –Z-side of the spacecraft bus and are canted 30° from the -Z axis, in the ±Y direction away from the X-axis. Because of the orientation of the RCS thrusters, the momentum unload maneuver produces a non-zero acceleration along the spacecraft –Z axis, generally in the anti-Sun direction.

Since the spacecraft attitude is entirely dependent on the science targets, accurate prediction of the momentum buildup on the spacecraft requires accurate prediction of the science targets. This is possible only by generating and following a planned series of observations, which the science community is not willing to commit to at this time. Initial project requirements listed a momentum unload frequency of no greater than once per day with a resultant velocity change (delta-V) of less than 1 meter per second (m/s). Subsequent analysis of the JWST spacecraft design by Northrop Grumman indicates that the most frequent momentum unload scenario occurs at a maximum roll angle with a single wheel failure. Under these circumstances, momentum unloads will be required every 4 days. Every

8 days is a more typical frequency for the momentum unloads. Depending on the axis of the momentum vector, the maximum delta-V will be up to 10 mm/s for each unload.

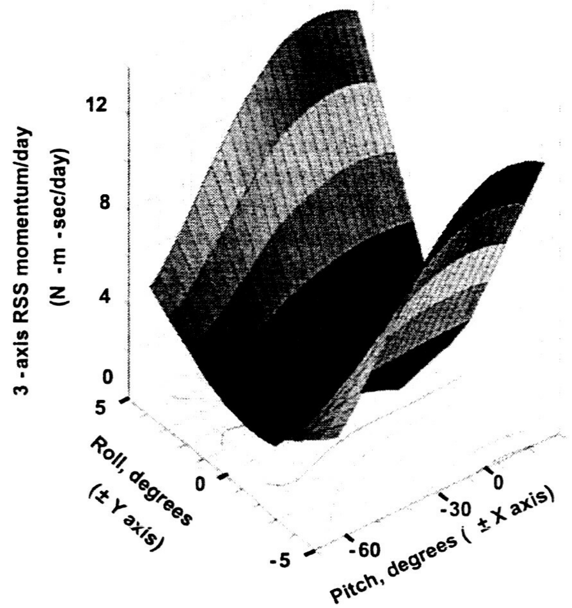

Figure 2. Daily Momentum Buildup (Courtesy of Northrop Grumman Space Technology)

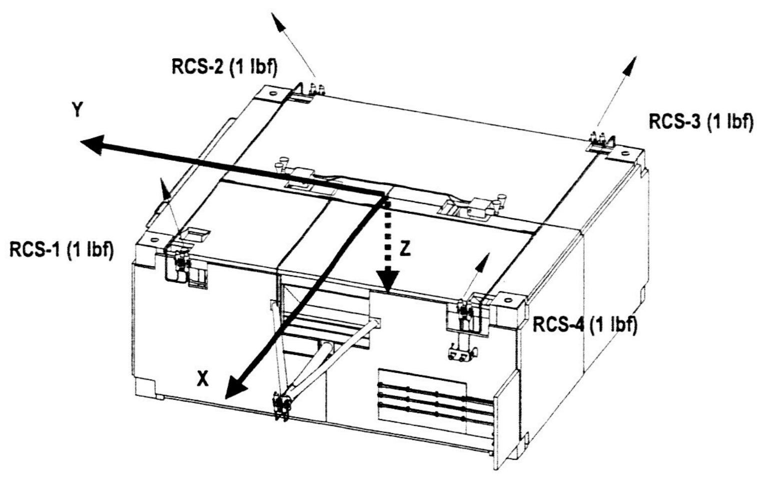

Figure 3. Spacecraft Bus and Thruster Configuration (Courtesy of Northrop Grumman Space Technology)

During a recent study, the possibility of adding two accelerometers was proposed. One linear accelerometer would measure the acceleration produced by the momentum unloads (on the order of 1 mm/s^2). The other would measure the acceleration due to the SRP force (on the order of $2x10^{-4}$ mm/s^2). Without accelerometer measurements, modeling of the momentum unload accelerations is limited by the accuracy of the propulsion system model, which is about 5%. Possible redesign decisions regarding thruster configuration and use of accelerometers are still under consideration.

3 – ORBIT DETERMINATION CONSIDERATIONS FOR JWST NAVIGATION

In this evaluation, the root-sum-square (RSS) position and velocity requirements for the JWST spacecraft were assumed to be on the order of 50 km and 20 mm/s (3-sigma). This velocity requirement was derived based on its contribution to the station-keeping delta-V budget. Navigation accuracy for orbits about the collinear Sun-Earth libration points is particularly sensitive to acceleration modeling errors because of the lack of observable dynamics. Ground station tracking has been used exclusively for all libration point missions to date. Range and Doppler measurements provide trajectory information content but only over a long data span, a minimum of about 21 days.

The presence of significant spacecraft acceleration modeling errors (e.g. SRP acceleration modeling errors due to uncertainties in the changing cross-sectional area and momentum unload delta-V modeling errors due to uncertainties in the magnitude and direction of the thrust) makes accurate orbit determination more difficult. The following are candidate approaches for handling the attitude reorientation and momentum unloads in the orbit determination process:

- *Do not model the acceleration changes*—Not modeling the changes in the SRP acceleration due to spacecraft reorientations between –30° and +30° introduces up to a 5% error in the SRP acceleration model for the JWST spacecraft; this approach was evaluated in this study. Not modeling the momentum unload delta-Vs introduces up to a 10 mm/s velocity error; this approach was not evaluated.
- *Model the acceleration changes*— Modeling the change in the cross-sectional area using attitude information available from the spacecraft can reduce the SRP acceleration modeling error to about 1%. Modeling the delta-V associated with the momentum unload based on the spacecraft attitude and thruster burn times available from the spacecraft can reduce the momentum unload modeling error to about 5%. These approaches, which were evaluated in this study, require data from the onboard attitude control and propulsion systems.
- *Estimate acceleration modeling corrections*—Estimation of the SRP coefficient used in the SRP acceleration model was evaluated. Estimation of momentum unload delta-Vs, which requires knowledge of the unload times, was not evaluated in this study.
- *Measure the acceleration changes*—Flying one or more accelerometers that continuously measure the spacecraft's SRP acceleration and the spacecraft's acceleration during the momentum unloads and including the acceleration measurements in the propagation of the spacecraft state vector was evaluated. For this approach, only the accelerometer measurement errors contribute to the navigation error.

4 – PRELIMINARY COVARIANCE ANALYSIS

A covariance analysis was performed to assess the sensitivity of the orbit determination accuracy to the frequency and magnitude of the momentum unloads. The Orbit Determination Error Analysis System (ODEAS) (ref. 1) was used to perform batch linear error analysis for several tracking and momentum unload scenarios. The tracking schedule consisted of the JWST baseline of Ground Network (GN) tracking for 30 minutes per day including both range and Doppler measurements, from three DSN sites. Previous analysis has shown that 21 days of measurement data are required for an observable solution using this schedule (ref. 2). An alternative tracking schedule of 4 hours every 10 days was considered but was not feasible with the current spacecraft design using a 21-day batch least-squares estimation time span. The error parameters used in this analysis, which are shown in Table 1, are consistent with expected 3-sigma error levels.

Simulations were made varying the frequency of momentum unloads from 1 to 8 days, where 1 day was the highest frequency in the original baseline design and where 4 days is the highest frequency and 8 days is a typical frequency in the current baseline. The delta-V from each unload was included in the trajectory propagation. The uncertainty in the delta-V magnitude was included as an error source for each simulation. Table 2 details the simulations performed and the estimated maximum RSS position and velocity uncertainties over a definitive time span of 21 days. As a reference point, the same simulations without any momentum unload delta-Vs produced a solution with a 11.5 km and 15.8 mm/s accuracy. The leading error sources in the covariance analysis were the SRP acceleration modeling error and the thrust modeling error for the momentum unloads. The assumed SRP acceleration modeling error was generally equivalent to about two momentum unloads with 5% uncertainties. For scenarios with three or more momentum unloads, the total error from the momentum unload errors dominated. For the scenarios with two momentum unloads, the total error contribution from SRP acceleration modeling errors and the momentum unloads delta-V errors were about equal. The range bias was the only other significant error source but its effect was much smaller than the SRP acceleration modeling or momentum unload errors.

Table 1. Covariance Analysis Error Parameters

Parameter	Value	Uncertainty
Area/mass ratio	0.03704	0
SRP coefficient	1.4	5%
Station location	DSN sites: DS16, DS46, DS66	3, 3, 3 m
Ionospheric refraction	DSN sites	100%
Tropospheric refraction	DSN sites	45%
Earth gravitational constant	Joint Gravity Model (JGM)-2	0.03 parts per million (ppm)
Earth non-spherical gravity	JGM-2 50x50	3*standard deviation
Sun gravitational constant	JGM-2	10 ppm
Moon gravitational constant	JGM-2	10 ppm
Range bias	GN ranging	15 m
Range noise	GN ranging	20 m
Doppler noise	GN ranging	8 mm/s

Table 2. Covariance Analysis Simulation Results

Unload Frequency (days)	Radial, Cross-track, In-Track Delta-V Magnitude (mm/s)	Delta-V Magnitude Uncertainty (%)	Maximum RSS Position Uncertainty (km)	Maximum RSS Velocity Uncertainty (mm/s)
no unloads			11.5	15.8
1	15, 15, 15	1.5	17.5	18.9
1	15, 15, 15	2	21.2	20.9
1	15, 15, 15	3	29.1	27.8
1	15, 15, 15	5	46.1	43.6
1	10, 3, 3	5	29.4	27.5
2	10, 3, 3	5	18.9	16.5
2	10, 3, 3	10	37.4	32.8
4	10, 3, 3	2	12.5	16.4
4	10, 3, 3	5	17.6	18.9
8	10, 3, 3	5	15.5	17.5

Figure 4 shows the estimated position and velocity accuracies versus unload magnitude uncertainty for a momentum unload frequency of one per day and delta-V magnitude of 15, 15, 15 mm/s. To meet the 20 mm/s velocity requirement with momentum unloads at one day intervals, the delta-V magnitude due to the unloads would have to be modeled to within 1.5%, which would require accelerometer measurements.

Reduction of the momentum unload frequency is critical to reducing the orbit determination uncertainty. Figure 5 shows the orbit determination accuracy achievable with various unload frequencies. Figure 5 assumes that the unloads are modeled to within 5%, which is consistent with the accuracy of a propulsion system model using finite burn models and telemetry data (e.g. spacecraft attitude and thruster burn times) with a delta-V magnitude of 10, 3, 3 mm/s. These results indicate that, with an unload frequency of 2 or more days, it may be possible to meet the orbit determination accuracy requirements with only a 5% requirement on the delta-V modeling accuracy (which does not require an accelerometer). However, past experience has shown that 3-sigma covariance analysis results for libration-point orbiters may be optimistic, giving error estimates that are consistent with the root-mean-square (RMS) overlap comparison differences in the operational solutions (ref. 3). Therefore, these results indicate that it is likely that an accelerometer will be needed.

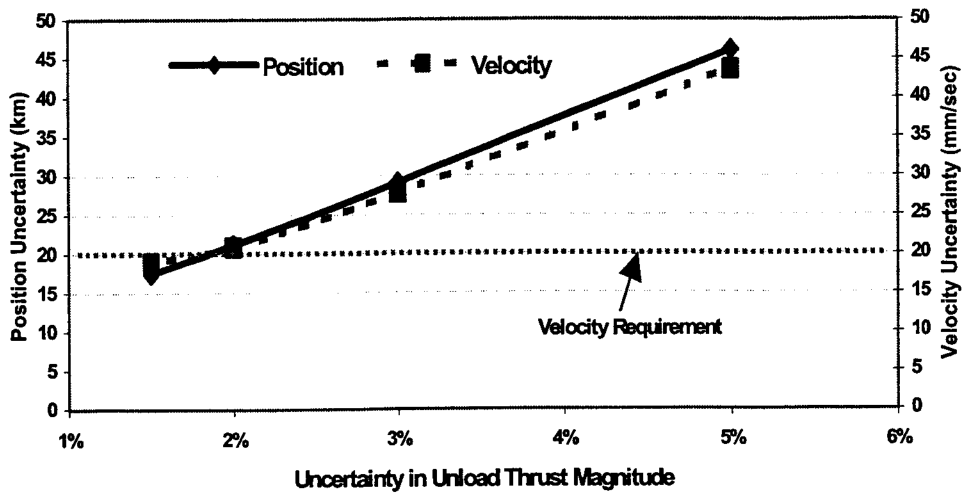

Figure 4. Effect of Delta-V Magnitude Uncertainty with Unloads at One-Day Intervals

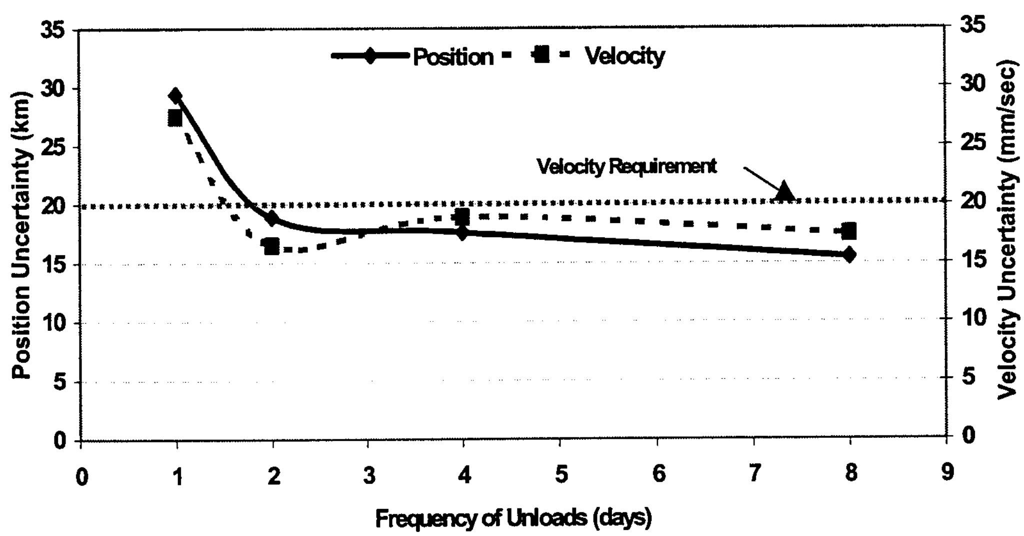

Figure 5. Effect of Unload Frequency on Orbit Determination Accuracy with a 5 % Delta-V Uncertainty

5 –NAVIGATION SIMULATION PROCEDURES

Measurements were simulated for the nominal JWST orbit using high-fidelity truth trajectories and realistic measurement noise and biases. The truth trajectories were generated using the Goddard Trajectory Determination System with the high-accuracy acceleration model listed in Table 3. There were 12 cross-sectional area changes and 11 momentum unload events in the 90-day simulation.

Table 3. Truth Trajectory Acceleration Model Parameters

Parameter	Values
Nonspherical Earth gravity model	4x4 JGM-2
Sun, Moon, Venus, Mars, Jupiter, Saturn positions	Definitive Ephemeris 200
Mean SRP coefficient	1.4
Satellite area model	Flat plate nominally perpendicular to the L2-Sun direction, with up to ± 30° attitude reorientation every 7 days
Maximum cross-sectional area (A)	200 meters^2
Mass	5400 kilograms
Momentum unload frequency	every 8 days
Momentum unload delta-V errors	• 0.12 mm/s • 1.2 mm/s

Attitude Reorientations

The spacecraft attitude reorientations were modeled as weekly changes with respect to the L2-to-Sun line. The primary angle α is defined so that when its value is zero, the Sun shield is perpendicular to the L2-to-Sun direction. The projected area toward the Sun is given by

$$A_p = A\cos(\alpha)$$

The angle α was varied between –30° and +30°, and the resulting area varied between 173 m^2 and 200 m^2. Not modeling this variation will produce ≤5% SRP acceleration modeling error. In the simulation, the assumption was made that the angle α can take on any value with equal probability. Figure 6 shows the projected area, A_p, as a function of the angle α, along with the area values that were modeled in the truth trajectory.

The position difference between a propagation using a constant mean cross-sectional area of 190 m^2 and a propagation using a cross-sectional area that changes every 7 days, varying between 173 m^2 and 200 m^2, grows to more than 150 km after 90 days, without including momentum unload delta-Vs. An accelerometer with an accuracy of 2×10^{-6} mm/s^2 (achievable with an ultra-precision accelerometer) would compensate for approximately 99% of the SRP acceleration error. The position difference between a propagation using cross-sectional areas with –1% to +1% errors and a propagation using the true cross-sectional areas that change every 7 days between 173 m^2 and 200 m^2 grows to about 15 km after 90 days.

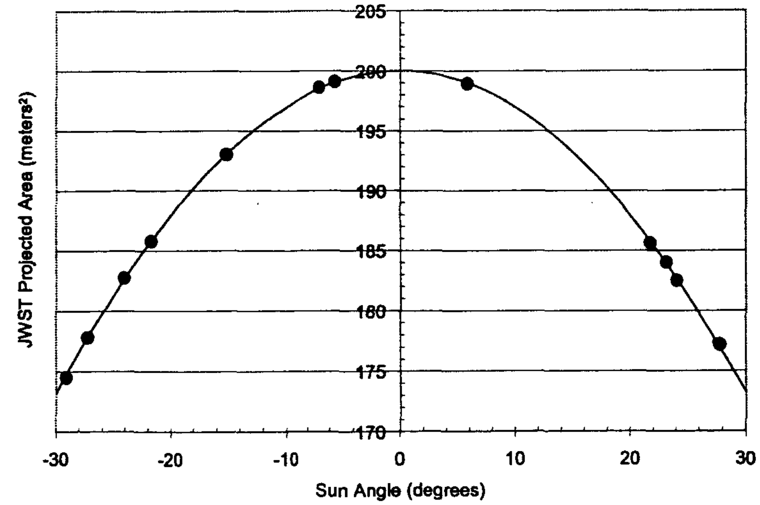

Figure 6. Projected Area Toward the Sun

Momentum Unload Delta-Vs

The momentum unload effects were modeled as impulsive delta-Vs, performed every 8 days, consistent with the expected unload frequency under normal conditions. The thrust plume is primarily Sunward. The momentum unload was modeled as follows in the Earth-Sun rotating frame, taking into account attitude reorientation effects:

$$\bar{\tau} = |\tau| \begin{pmatrix} \cos(\alpha) \\ \sin(\alpha) \\ 0 \end{pmatrix}$$

The magnitude of each momentum unload burn, $|\tau|$, will be less than 10 mm/s. Assuming a 1 pound thruster with a 5400 kilogram spacecraft, a 10 mm/s delta-V will take up to 12 seconds. In this case, if the momentum unload accelerometer has an accuracy of 0.1 mm/s^2, a delta-V error of 1.2 mm/s would be created for a 12 second burn.

The position difference between an orbit propagation using cross-sectional areas with ±1% errors and RSS momentum unload delta-V errors of 1.2 mm/s and an orbit propagation using true cross-sectional areas that change every 7 days between 173 m^2 and 200 m^2 without delta-Vs grows to about 60 km after 90 days. When the momentum unload delta-V errors are reduced to 0.12 mm/s, the propagation error reduces to about 17 km after 90 days. These propagations indicate that a 5% SRP acceleration modeling error is more significant than a 1.2 mm/s momentum unload delta-V error, which is more significant than a 1% SRP acceleration modeling error, which is more significant than a 0.12 mm/s unload delta-V error.

5.1 –Ground Station Measurement Simulation

Table 4 lists the ground station measurement simulation parameters. Standard two-way range and S-band Doppler measurements were simulated using the nominal tracking schedule of one measurement pair every 10 seconds for one 30-minute contact every day and an alternate schedule of every 10 seconds for one 4-hour contact every 10 days. Ground-to-spacecraft one-way Doppler measurements were simulated every 10 seconds for one 4-hour contact every 10 days. Ground-to-spacecraft one-way Doppler measurements are derived from the standard S-band communications signals transmitted by the DSN ground stations using a Doppler measurement capability implemented within the communications receiver onboard the spacecraft (ref. 4). The assumption was made that the spacecraft time is determined to better than 1 microsecond independent of the onboard Doppler measurement capability. The one-way Doppler measurement accuracy is primarily dependent on the noise and stability characteristics of the onboard oscillator that provides the frequency reference used in the Doppler measurement process. The baseline reference frequency quality was modeled based on a high-quality ultra-stable oscillator (USO) with a drift rate of 0.02 Hertz-S per day.

Table 4. Ground Station Measurement Simulation Parameters

Parameter	Nominal Value
Ground tracking station	Goldstone (DSN)
Ground tracking schedule	1 measurement pair every 10 seconds for one 30-minute contact every day
Ground tracking visibility constraints	10° elevation angle
Ground tracking measurement types/random errors	Range: 2 meters Doppler: 0.001 Hertz-S (5E-13 parts)
Ground-to-spacecraft atmospheric errors	0.1 Hertz-S (maximum)
USO frequency drift rate	Two-way Doppler: 0 One-way Doppler: High quality USO: 0.02 Hertz-S per day (1E-11 parts per day)
Initial receiver frequency bias	0 Hertz

Figures 7 and 8 show the magnitude of the simulated ground-to-spacecraft range and Doppler measurements for the daily tracking schedule, respectively. The range measurements are nearly constant during each contact and change slowly over 90 days (one-half the orbital period). In this example, the range is increasing because the spacecraft is moving from the point of closest approach to the Earth to the far side of L2 from the Earth. The Doppler measurements change by about 300 Hertz-S during each 30-minute contact due to the Earth's rotation and change slowly over 90 days. Figure 8 also shows the effect of the Moon's gravity on the Doppler measurements, which produces a 28-day oscillation in the curve. The Doppler values reach their maximum when the spacecraft is at the maximum ecliptic inclination, i.e. farthest out of the Sun-Earth plane. The range and Doppler curves will mirror the first 90 days' behavior as the spacecraft traverses the second half of the libration point orbit.

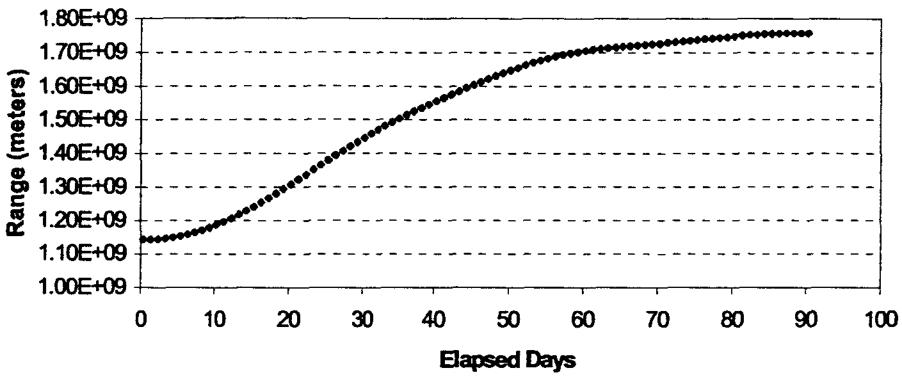

Figure 7. Simulated Ground-to-Spacecraft Range Measurements (one contact every day)

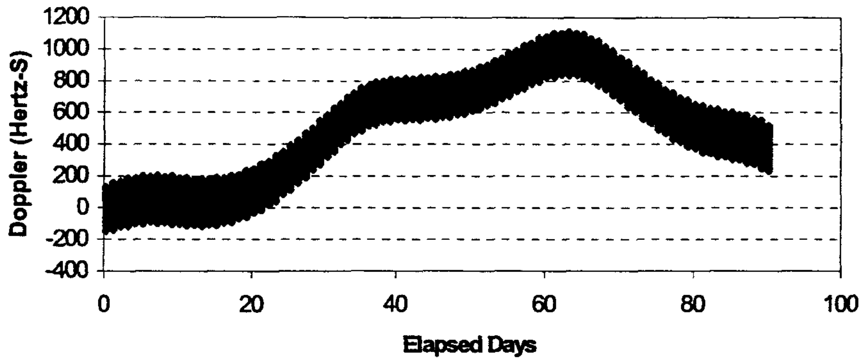

Figure 8. Simulated Ground-to-Spacecraft Doppler Measurements (one contact every day)

5.2 Celestial Object Measurement Simulation

The celestial object measurements, shown in Figure 9, consist of either line-of-sight (LOS) unit vectors from a spacecraft sensor to a near body (e.g. Sun, Moon, or Earth) measured in the spacecraft body frame, star-to-near-body pseudoangle (PA) measurements, or near-body-to-near-body pseudoangle measurements. The celestial object navigation concept was motivated by the celestial position fix discussion presented in reference 5. The pseudoangle measurements are equal to the cosine of the angle between the LOS vectors to a near body and to either a star or another near body. Pseudoangle measurements are independent of the reference frame and, therefore, do not require knowledge of the real-time attitude solution. The detailed algorithms are defined in reference 6.

The baseline celestial object measurement noise and bias characteristics listed in Table 7 are based on the following assumptions:

- For the spacecraft-to-Sun LOS measurements, the sensor noise and bias characteristics are consistent with the performance of the Adcole 60 arc second digital Sun sensor flown on the Chandra spacecraft (i.e., 5 arc seconds noise and 40 arc seconds bias). The onboard attitude determination errors for JWST are expected to be about 5 arc seconds before guide star acquisition with a 1 arc second bias.

- The pseudoangle measurements are obtained using a single optical sensor with measurement noise and bias characteristics consistent with the current state-of-the-art optical technology (i.e., ≤1 arc second noise, bias error equal to one tenth of the noise).

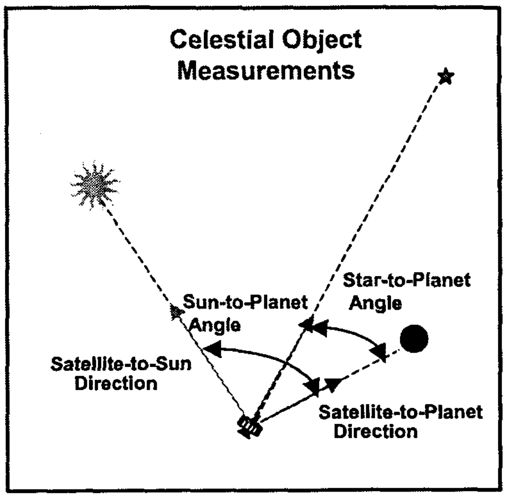

Figure 9. Celestial Object Measurements

Table 7. Celestial Object Measurement Simulation Parameters

Parameter	Nominal Value
Celestial object measurement types	LOS to Sun Earth-to-Moon PA Northern star-to-Moon PA
Celestial object measurement rate	Every 60 seconds from each sensor
Random LOS measurement errors (Sun sensor + attitude noise)	8 arc seconds = 3.8E-5 parts
Sun sensor measurement biases (sensor local + attitude-error-related misalignments)	40 arc seconds = 1.9E-4 parts
Random pseudoangle measurement errors	1 arc second =4.8E-6 parts
Pseudoangle measurement biases	0.1 arc second =4.8E-7 parts

5.3 Measurement Processing

The extended Kalman filter algorithm available in NASA GSFC's GPS Enhanced Onboard Navigation System (GEONS) flight software was used to process these measurement sets (ref. 6). The filter's velocity process noise variance rate was adjusted to accommodate the acceleration modeling uncertainties. The absolute navigation errors were computed by differencing the truth and estimated state vectors.

6 –GROUND NAVIGATION FILTER PERFORMANCE

The accuracy of the ground navigation approach was evaluated using DSN tracking schedules of one 30-minute contact every day and one 4-hour contact every 10 days. The impact of the attitude reorientations and momentum unloads on navigation accuracy was assessed. The tracking schedules yield one range and Doppler measurement every 10 seconds during each contact from the DSN station at Goldstone. Note that these simulations used 1-sigma error levels and therefore the results reflect 1-sigma performance expectations. The maximum position and velocity steady-state errors are summarized in Figure 10. The associated RMS errors are between 50 and 65 percent of the maximum errors. Figures 11 and 12 provide representative results for the error in the estimated state (solid line) and estimated state root-variance (dashed line).

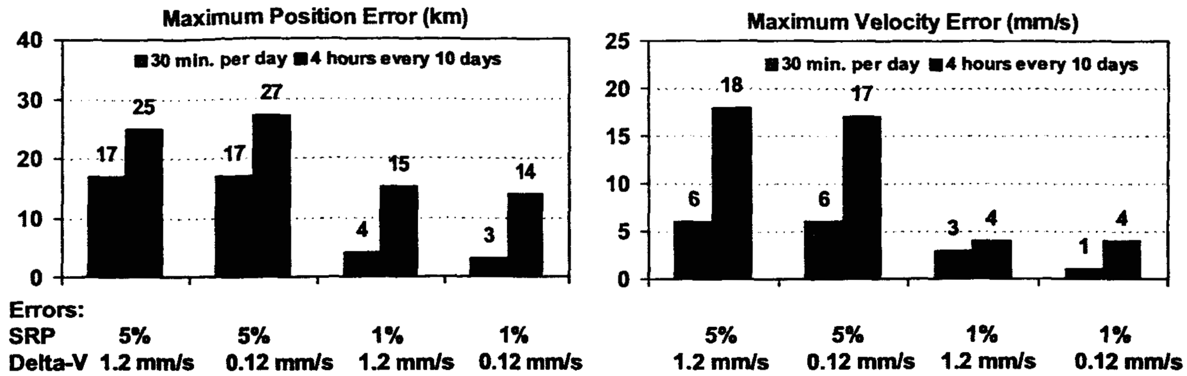

Figure 10. JWST Ground Navigation Steady-State Performance

Significantly more accurate and more stable solutions were achieved with daily tracking contacts than with only one contact every 10 days. With daily tracking, the performance was much less sensitive to filter tuning parameters, indicative of a more stable solution. With one tracking contact per day, steady-state performance was achieved after about 20 days of processing and the maximum steady-state position and velocity errors were well below the maximum 50 km and 20 mm/s requirements for JWST for all cases.

When attitude reorientations were included in the truth simulation but not modeled in GEONS, the resulting SRP acceleration modeling errors (up to 5%) were the dominant error source regardless of whether the momentum unload modeling errors were 0.12 mm/s or 1.2 mm/s (Figures 11 and 12). With daily tracking, better velocity solutions were obtained by estimating a correction to the SRP coefficient (Figure 11). With proper tuning, accurate estimation of the SRP correction was achieved within 3 days of the attitude reorientation. Reducing the SRP acceleration modeling errors to 1% (either by modeling the reorientation changes or including SRP acceleration measurements) significantly improved the solution accuracy with either 0.12 mm/s or 1.2 mm/s momentum unload modeling errors (Figure 13). When momentum unload modeling errors were significant (i.e. ≥ 1.2 mm/s), their impact on velocity errors was mitigated in the case of daily tracking by incrementing the velocity variances at the unload times.

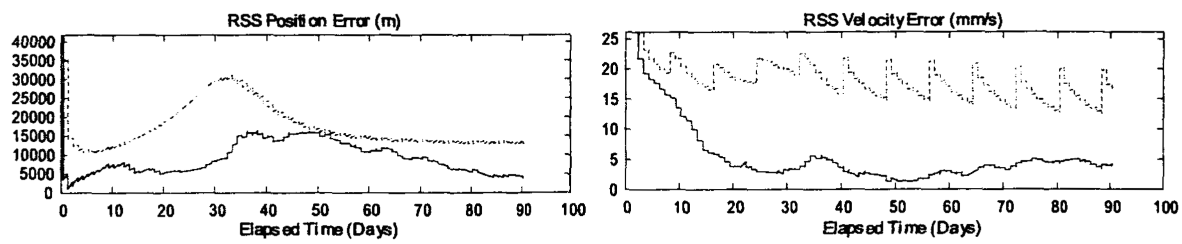

Figure 11. Range and Doppler Measurements with 5% SRP and 1.2 mm/s Delta-V Modeling Errors (1 contact every day)

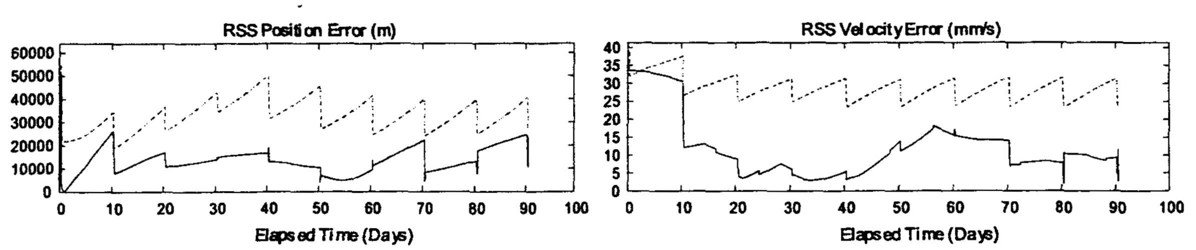

Figure 12. Range and Doppler Measurements with 5% SRP and 1.2 mm/s Delta-V Modeling Errors (1 contact every 10 days)

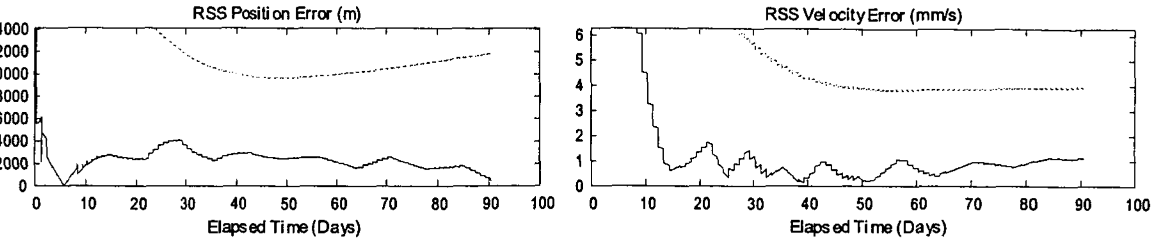

Figure 13. Range and Doppler Measurements with 1% SRP and 0.12 mm/s Delta-V Modeling Errors (1 contact every day)

7 –ONBOARD NAVIGATION FILTER PERFORMANCE RESULTS

The accuracy of onboard navigation was evaluated with measurements from only the sensors currently planned for the JWST and using pseudoangle measurements from additional optical navigation sensors. The impact of the attitude reorientations and momentum unloads on navigation accuracy was investigated for each of these scenarios.

7.1 Onboard Navigation using Baseline Sensors

The baseline sensors consist of a radio frequency receiver augmented with the capability to measure the ground-to-satellite Doppler shift and a Sun sensor that measures the sensor-to-Sun LOS vector. The tracking schedule produced one ground-to-spacecraft one-way Doppler measurement every 10 seconds during each 4-hour contact every 10 days from Goldstone. The Sun sensor measurements were processed at a rate of one per minute.

Solutions were obtained processing only one-way Doppler measurements with 1% SRP acceleration modeling errors and 0.12 mm/s momentum unload delta-V errors (Figure 14). For these solutions, the maximum errors of 130 km and 75 mm/s are significantly larger than the JWST requirements. Accurate estimation of the frequency bias was found to be very sensitive to the value of the frequency bias process noise variance rate, indicative of an unstable solution. In addition, the covariance estimates do not indicate that the filter is convergent. Increasing the tracking contacts to one 30-minute contact per day for the first 30 days followed by one 30-minute contact every 3 days did not significantly improve navigation performance. The addition of the sensor-to-Sun LOS measurements consistent with the Sun sensor baselined for JWST (8 arc second noise, 40 arc second bias) did not improve the position and velocity solutions as compared to the one-way Doppler-only solution.

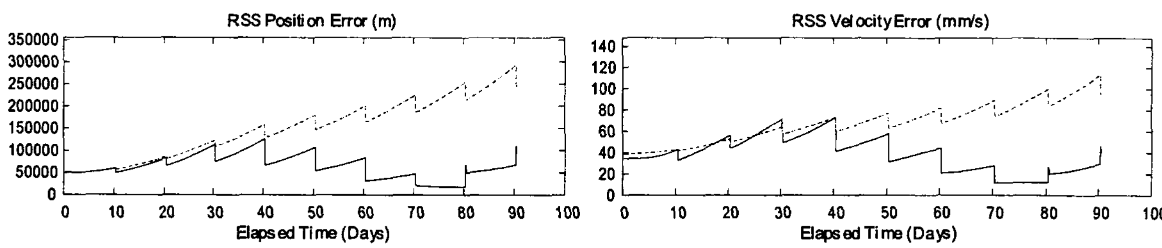

Figure 14. One-Way Doppler Measurements with 1% SRP and 0.12 mm/s Delta-V Modeling Errors (1 contact every 10 days)

7.2 Onboard Navigation using Celestial Navigation Sensors

Solutions were obtained processing optical-quality pseudoangle measurements (1 arc second noise, 0.1 arc second bias) with and without one-way Doppler measurements. The pseudoangle measurements were processed at a rate of one per minute per sensor. Figure 15 summarizes the maximum position and velocity errors for each steady-state solution. The associated RMS errors are between 50 and 65 percent of the maximum errors.

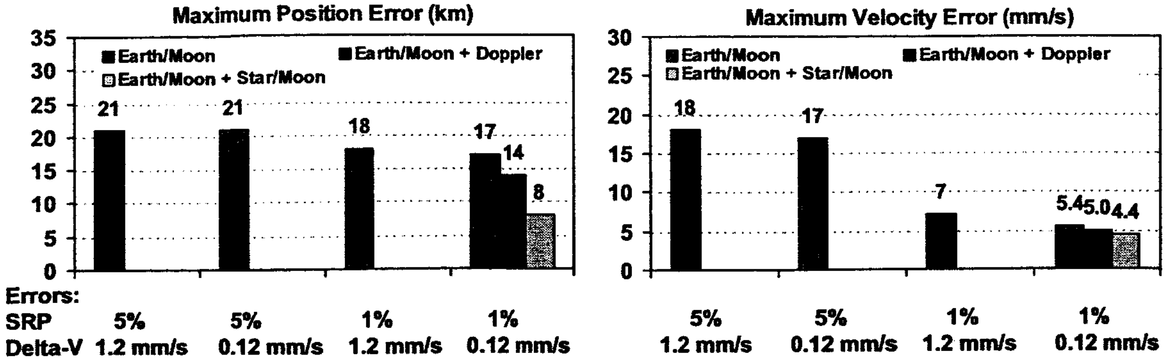

Figure 15. JWST Onboard Navigation Steady-State Performance

Solutions were obtained processing only Earth/Moon pseudoangle measurements, varying the magnitude of the acceleration modeling errors. When attitude reorientations were modeled in the truth trajectory but not included in GEONS (producing up to a 5% SRP acceleration modeling error), maximum errors of about 21 km and 18 mm/s were obtained for cases with 0.12 mm/s and 1.2 mm/s momentum unload modeling errors (Figure 16), respectively. Reducing the SRP acceleration modeling errors to 1% (either by modeling the reorientations or including accelerometer measurements) significantly reduced the velocity errors to less than 8 mm/s (Figure 17). In the case of 1% SRP errors, estimation of the SRP coefficient correction did not improve the solution.

The addition of realistic one-way Doppler measurements produced about a 20% reduction in position errors as compared with solutions using only Earth/Moon pseudoangle measurements. The addition of Northern star/Moon pseudoangles reduced filter convergence time to about 15 days and reduced the steady state maximum errors to 7.6 km and 4.4 mm/s with 1% SRP acceleration modeling errors and 0.12 mm/s momentum unload modeling errors (Figure 18).

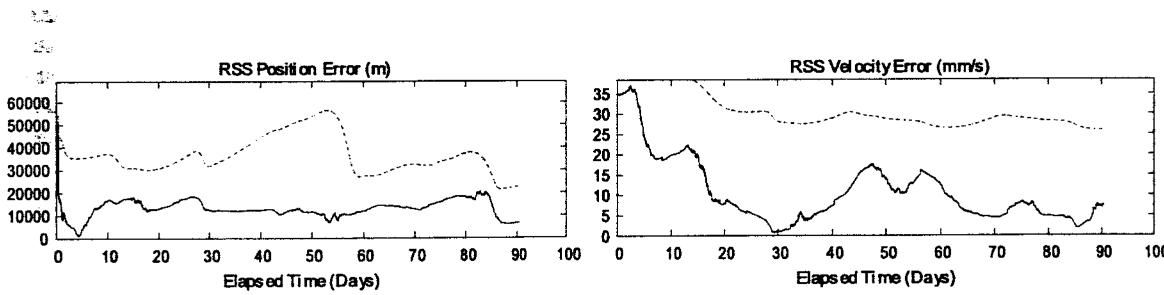

Figure 16. Earth/Moon Angles with 5 % SRP and 1.2 mm/s Delta-V Modeling Errors

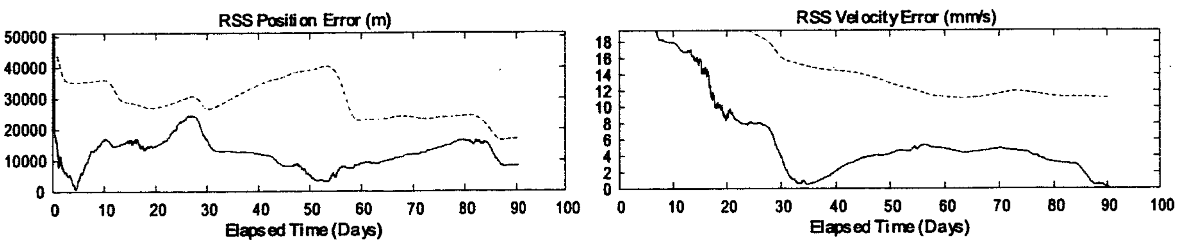

Figure 17. Earth/Moon Angles with 1 % SRP and 0.12 mm/s Delta-V Modeling Errors

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ACRONYMS

DSN	Deep Space Network
FOR	field of regard
GEONS	GPS Enhanced Onboard Navigation System
GN	Ground Network
GPS	Global Positioning System
GSFC	Goddard Space Flight Center
JGM	Joint Gravity Model
JWST	James Webb Space Telescope
km	kilometer
lbf	pound
LOS	line-of-sight
m	meter
mm	millimeter
mm/s	millimeter per second
N	Newton
NASA	National Aeronautics and Space Administration
ODEAS	Orbit Determination Error Analysis System
PA	pseudoangle
ppm	parts per million
RCS	Reaction Control System
RMS	root-mean-square
RSS	root-sum-square
s	second
SRP	solar radiation pressure
USO	ultra-stable oscillator